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Abstract

Accurate simulation of radiative heat-transfer effects from the rocket engine exhaust plays an important role for the proper characterization of missile base heat loads. To promote improved radiative transfer solutions, careful attention to the physical flow-field models is paramount. Use of a generalized fluid dynamic model can assist in the close approximation of the actual base heating by solving the fully coupled, two-phase, chemically reactive, Navier-Stokes equations in multiple dimensions. Solutions to this set of governing equations enables flow simulations for the complex expansion of the fuel-rich engine exhaust gases. Some key features for these expansion processes include phenomena such as baseflow recirculation and separation, atmospheric entrainment, and shock structures that result from interactions with the vehicle and the natural expansion of the plume flow field into the quiescent environment. Three-dimensional aspects of the reacting gas dynamic flow processes are also very important components, especially in the missile base and the near engine exhaust regions.

A computer model called GPACT (**G**eneral **P**ropulsion **A**nalysis **C**hemical Kinetic and **T**wo-Phase) includes numerical approximations for these physical processes, and is currently under development. GPACT was previously applied to simulate the Titan II flow field at 46 km, in its entirety, and to model the flow field of a subscale liquid-propellant rocket engine (LRE) missile fired at 10.1 km in a ground test environment. The ability of this flow-field model to simulate physical details of the flow processes contributing to the radiative heating will be presented in this paper. A variety of flow-field model approximations are examined in order to isolate the influences of three dimensionality and upstream solid boundary effects on the calculations. Radiative heat-transfer solutions are obtained for several flow-field examples in order to exemplify the importance of flow-field approximations on the radiation component of the overall base heat loads.

Background

GPACT (**G**eneral **P**ropulsion **A**nalysis **C**hemical Kinetic and **T**wo-Phase) is a computer program currently under development that has been applied towards simulating propulsion-generated flow-field phenomena. These simulations treated fully-coupled, three-dimensional flows with finite-rate chemistry. This capability was described and the results of validation studies are reported in previous studies.[1,2] The adequacy of this flow-field model to simulate details of the physical phenomena contributing to the IR (Infrared) radiant heating will be indirectly assessed in this study. GPACT was derived from a research version of the Generalized Implicit Flow Solver, GIFS. GIFS was originally developed by Holcom in the mid-80s.[3] Since its inception, the GIFS computer program has been systematically and extensively modified under the joint sponsorship of the Air Force Research Laboratory and the Arnold Engineering Development Center.[1-4] These changes have significantly improved the robustness, generality of the solution algorithm, the physical model approximations, and internal databases. This has led to the evolution of the GPACT computer model.

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Recent efforts have been directed towards improving the overall computational efficiency, including turn-around time and computer memory requirements. Further, modification of the solution technique to strengthen the coupling of numerous physical phenomena occurring in rocket propulsion flow fields that focus on chemical kinetics and two-phase flow with phase change are also being addressed, since these phenomena can significantly impact radiant emission.

Previous work documented several propulsion flow-field simulations which assess the effects of various flow phenomena, including 3D missile/body/base geometry, chemical kinetics, and turbulence.[2] The process of refining these solutions identified needed computer program enhancements that have culminated in the current status of the GPACT model development. This development has been guided by results of numerous comparisons of simulated propulsion flow-field phenomena with observations of propulsion flight systems and a validation database of laboratory measurements focusing on specific isolated flow-field phenomena and simplifying assumptions such as three dimensionality, turbulent mixing, and kinetic rate-controlled chemistry.[1]

The vision guiding future GPACT computer program enhancements is to extend the development of the model beyond the "research" level and eventually provide an applications-oriented, CFD flow-field simulation tool for use by CFD users in the propulsion community. In order to accomplish this goal, considerable work remains to be completed. This effort represents a beginning toward applying the GPACT model for detailed propulsion flow-field simulations, and initiates verification and validation of the model's results using radiative heat transfer as an evaluation parameter. Use of flow-field radiance as a discriminant is due to the sensitivity of IR radiation to the details of the gas dynamic and chemical flow-field properties. Shock structure, changes to the composition state of the mixture due to chemical reactions, and state of the particle properties can readily be inferred from the radiant flow-field emission.

Titan II Computations

This paper utilizes previously reported flow-field results from Refs. 2 and 4 and evaluates the coupled effects of chemical kinetics, missile body/plume interactions, and mixing on the calculated IR radiative heat transfer. A gaseous band model formulation of the IR radiative transport equations[5] is applied in an uncoupled fashion to produce the in-band IR radiation properties. Spatial flow-field properties (pressure, temperature, and chemical species concentrations) from the GPACT model computations are input to the radiative transfer model to determine in-band (a) spatial radiant intensity and (b) axial emission profiles (aka station radiation). The radiative transfer calculations from the flow-field solutions are compared to assess the sensitivity of the predicted IR radiative heat transfer to various simplifying flow-field assumptions.

Parametric flow-field simulations from the Titan II dual-nozzle propulsion system operating at high-altitude flight conditions (47 km) are reported in Ref. 2. This high-altitude flow field is dominated by a distinct inviscid plume exhaust structure that includes the plume barrel shock (expansion shock) and the barrel shock reflection (regular reflection) at the plume centerline. Radiative heat-transfer prediction assessments were not included in the high-altitude computational study described in Ref. 2; however, simulations of the Titan II SLV flow fields were completed to assess the effects of three dimensionality, missile body/plume interactions, gas generator exhaust, and reacting flow approximations on the simulated plume exhaust properties. The current work examines infrared radiation heat-transfer calculations for representative Titan II flow-field simulations. The Titan II is a dual-engine system with nearly identical nozzle geometries and operating conditions. The inflow conditions for the simulations are given in Table 1.

The key geometric features used within this study of the Titan II vehicle include the aft portion of the missile body region, the axisymmetric dual nozzles, and the gas generator hardware. Details are schematically shown in Fig. 1. Flow-field and IR radiant heat-transfer calculations were performed

Table 1. Input Conditions for the 47-km Titan II Computations

Free-stream Conditions at 47 km
- Temperature = 269 K
- Axial Velocity = 1877.6 m/sec (572 ft/sec)
- Static Pressure = 0.001 atm (0.0146 psia)
- Mach Number = 5.7
- Species Concentrations (Mass Fraction)

$N_2 = 0.77$
$O_2 = 0.23$

Nozzle Exit Conditions (One Dimensional, Both Nozzles)
- Static Temperature = 1,920 K
- Axial Velocity = 2776.6 m/sec (846 ft/sec)
- Radial Velocity = 0 ft/sec
- Static Pressure = 0.915 atm (13.45 psia)
- Mach Number = 3.0
- Nozzle Exhaust Species Concentrations (Mass Fraction)

CO = 0.039	CO_2 = 0.1811	H_2O = 0.3496	
N_2 = 0.414	NO = 0.0109	OH = 2.139e-3	
H_2 = 3.13e-3	H = 1.24e-4	O_2 = 0.0	O = 0.0

assuming an axisymmetric, single equivalent engine approximation with no vehicle body included in the computational domain. For the cases without the vehicle geometry, effects are also known as the plume-only or "flying plume" approach. The axisymmetric solution is compared to a more complete 3D physical simulation, including the flow field surrounding the missile body and base region. Calculated temperature contours and axial centerline profiles, for cases with and without the missile body/base, are shown in Fig. 2.

When the missile body is included in the calculation, more intense combustion occurs in the near-field shear layer. Persistent shear layer heating is most evident in the "flying" plume (no vehicle effects) result. This heating is primarily caused by the velocity differential of the two mixing streams (i.e., viscous heating) and is not the result of combustion due to atmospheric entrainment processes in the far field (i.e., afterburning). Calculated axial profiles of the centerline static temperature, with and without the missile body, indicate that the shock reflection point is located approximately 12 m further downstream when the missile body is included. A close-up view of the static temperature contours in the missile base flow region is shown in Fig. 3. This flow includes the gas generator effluent. The subsequent heating of the missile base surface and the effect of the gas generator exhaust gases impinging on the nozzle surfaces are clearly evident. The gas generator flow initially expands as it exits the nozzle and adjusts to the ambient pressure condition. Further downstream in the region between the two nozzles, the gas generator flow area is compressed because of the flow area change created by the nozzle expansion skirts. A hot, high-pressure region is created at this location.

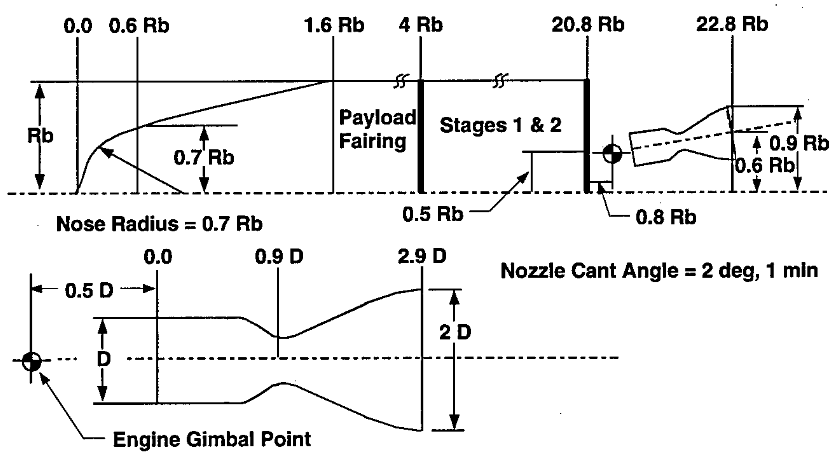


Fig. 1. Vehicle dimensions.

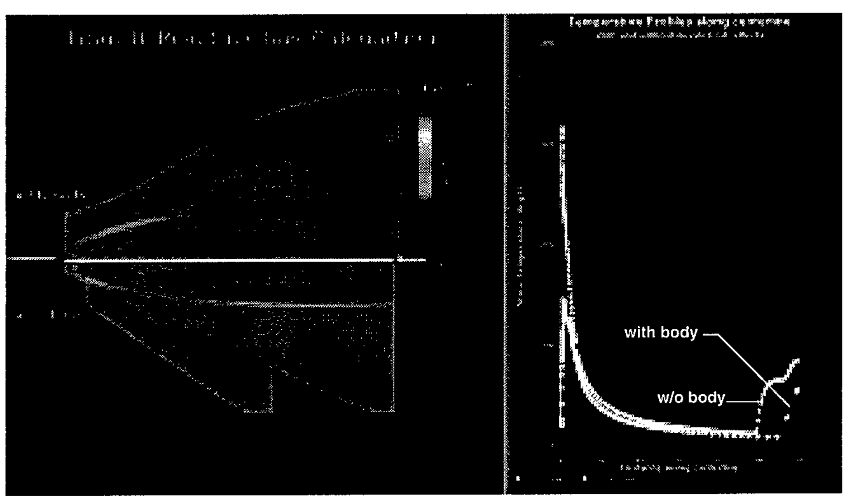


Fig. 2. Static temperature contours with and without missile body.

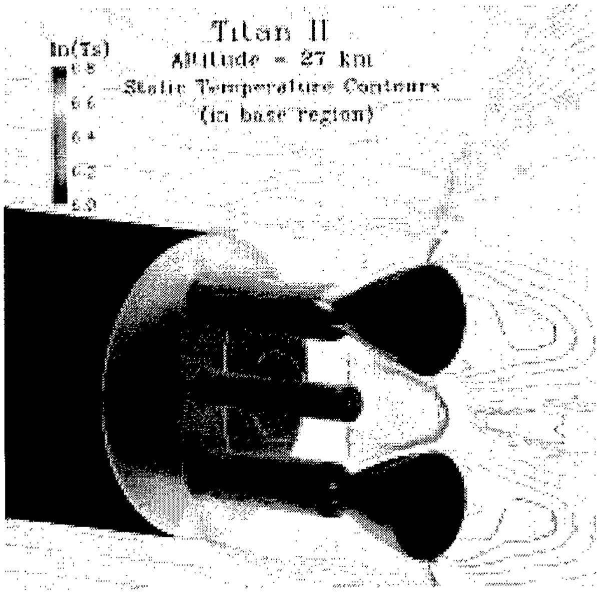


Fig. 3. Titan II SLV at 47.6 km reacting flow calculation near-field results.

Corresponding short wave IR (SWIR) image simulations for the "flying" plume and the 3D body/plume approximations are shown in Fig. 4. These results indicate that the effects of the missile body/base region significantly increase the intensity levels in the far-field regions of the plume exhaust and also increase the plume width throughout the computational domain. The effects of plume/atmosphere mixing and subsequent burning also are evident in these results.

Integrating the radial components of the IR image results in an axial emission profile, also referred to as station radiation profile. The usefulness in this representation of the IR emission is that it can more quantitatively compare two solutions, as well as depict relative trends in far-field emission mechanisms, such as the location of the barrel shock reflection point and an indicator of relative afterburning conditions. In-band station radiation for this case is shown in Fig. 5. Station radiation was calculated for the 3D body/plume simulation and the axisymmetric, single equivalent nozzle (SEN) approximation. The SEN approximation conserves the mass, energy, and momentum of the dual-nozzle configuration by modifying the area ratio and throat area of a single axisymmetric nozzle.

For these cases, the primary impact on the radiative heat-transfer characteristics comes from the 3D treatment of the flow. This primarily impacts the shape of the near-field flow, causing the 3D flow

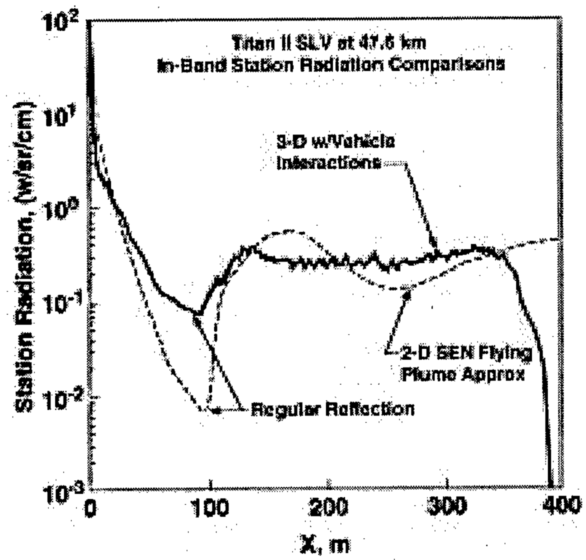


Fig. 5. Comparisons in the SWIR station radiation profiles between 3-D body plume and 2-D axisymmetric flying plume.

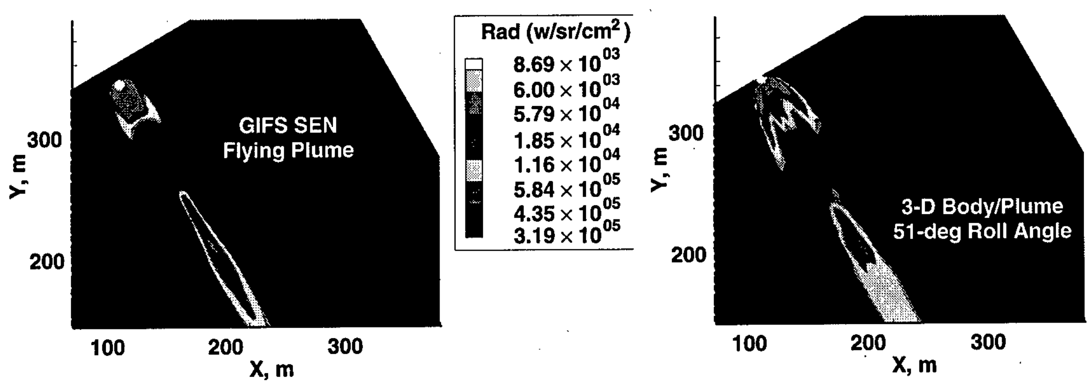


Fig. 4. Focal plane array images for the SWIR depicting the difference in predicted spatial radiance distribution with three meter pixel resolution.

field to appear much more extended than the axisymmetric flow-field treatment. In terms of the SWIR radiance features, the 3D solution presents a larger near-field cross section than does the SEN image. In the far field, the SEN result appears to be influenced by gas dynamic wave oscillations from the multiple reflections of the barrel shock. The primary gas dynamic feature that causes much of the differences seen in these comparisons comes from the intraplume jet that is produced when the dual motor plumes impinge upon each other.

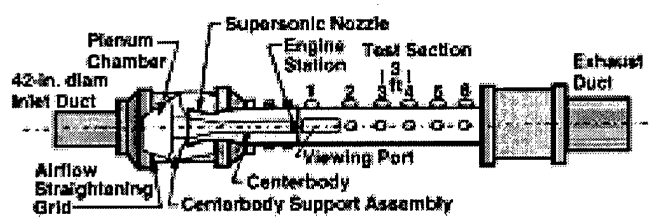
Subscale Test Computations

For this effort, GPACT simulations of the flowfield environment for a subscale, multiple nozzle, liquid-propellant rocket engine operating in a ground test chamber (referred to as T6) at 10.1 km were also completed.[6] The low-altitude condition results in significant plume/atmospheric combustion (afterburning). This phenomenon was confirmed by measurements, and also captured by the GPACT model simulations. In-band plume IR radiative properties resulting from the flow-field calculations will be simulated to assess the influence of the 3D, missile body/base geometry approximations and chemical kinetics.

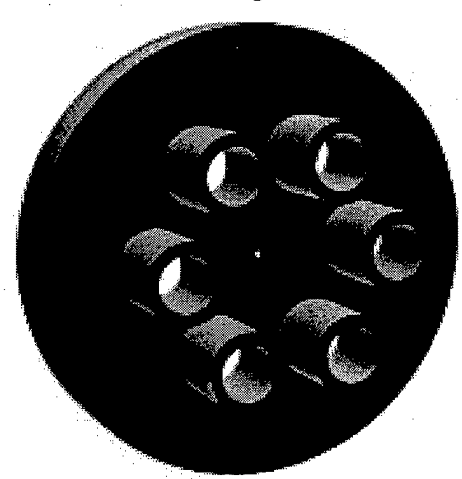
The results from these calculations indicate that 3-D geometries, including the missile body/base, and chemical kinetics play a significant role in affecting the plume IR radiative transfer. Global flow-field sensitivity trends which were previously reported[2] are magnified in the IR radiative transfer assessment. The flow-field calculations clearly identified the significance of 3D missile body/base influences and chemistry on the global structure of the spatial flow-field properties. The insight gained from the evaluation of the calculated gas dynamic and chemical flow-field phenomena is a tremendous aid in understanding the IR simulation results.

Parametric flow-field studies focused on the effects of simplifying assumptions which are often made due to the limitations in modeling techniques, computational resources, and time. Numerical simplifications often involve approximating multiple engines and other 3D effects (e.g., angle of attack, fins, steering vanes, turbine exhaust ports, etc.) using axisymmetic assumptions. These assumptions often compromise the complexities present in true 3D flows, such as the complex recirculation zone in the missile base region, or multiple engine exhaust interactions (i.e., plume/plume inpingement). Chemistry effects can also be compromised by extreme bounding conditions assuming frozen or equilibrium approximations. For some applications, these simplifying assumptions might be adequate. However, the stringent accuracy requirements needed for radiative heat-transfer analyses, especially in the base region, necessitate numerical representation of 3D effects and chemical kinetics for liquid propulsion systems. This is especially true if the flow conditions are conducive to plume/atmosphere combustion and the formation of gas dynamic wave structure and embedded combustion zones in the downstream region.

The IR radiation resulting from an intensely afterburning subscale liquid-propellant rocket engine tested in the T-6 Experimental Development Test Cell Facility at the Arnold Engineering Development Center (AEDC) was simulated and compared with IR in-band station radiation measurements.[6] A schematic of the T6 test cell config-



a. Test configurations



b. Computational configuration

Fig. 6. AEDC plume intelligence test configuration.

uration is shown in Fig. 6a. Viewing ports located axially along the test section were used to obtain in-band station radiation measurements. The viewing ports are approximately 3 ft apart and allow for station radiation measurements beginning at the nozzle exit plane extending axially to 18 ft. The subscale test article consisted of six identical nozzles configured as shown in Fig. 6b. The nozzle exit diameter is 1.1 in., and the diameter of the circular base region supporting the six nozzle cluster is 35 in. The total pressure and temperature in the combustion chamber was 500 psia and 3,000 K, respectively. The nozzle exit Mach number was 2.9, and the area ratio of each nozzle was six.

Normalized in-band station radiation predictions resulting from a SEN approximation of the six nozzle cluster (i.e., no body flying plume), and the complete geometry, including missile/body/base, and "flying" plumes results are shown in Fig. 7. Normalized measurements obtained during the test are also included for qualitative evaluation. The low-altitude test environment and fuel-rich plume exhaust creates intense afterburning that overshadows any gas dynamic wave structure effects in the IR radiation simulation. The SEN approximation and complete geometry results shown in Fig. 7 indicate an intense afterburning region initiated downstream of the nozzle exit, peaking in intensity at approximately 1 ft downstream. However, in the SEN results, the afterburning quickly subsides and the corresponding intensity levels immediately decrease and remain at a lower level. This is not consistent with the trends observed in the measurements. The data also indicate that afterburning is initiated approximately 1 ft downstream of the nozzle exit, but the peak intensity level is maintained throughout the first 4-5 ft. The complete geometry simulation agrees qualitatively with the measurement results and appears to capture the observed phenomenology. The major differences observed in the simulation comparisons are attributed to the absence of flameholding effects in the SEN approximations which are attributed to the presence of the missile base. As shown in the complete geometry simulation, the missile base region acts as a flameholder and maintains the afterburning intensity to greater axial extents.

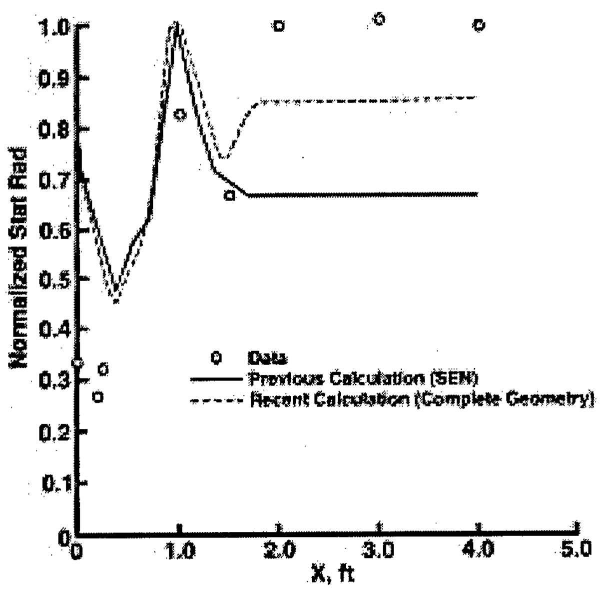


Fig. 7. Comparison of GIFS and data for baseline in-band radiation for CO CO_2 band.

Conclusions

These calculations indicate that three-dimensional flow is an important factor for the simulated cases that were examined in this study, and can substantially influence the accuracy and interpretation of the simulated radiative heat-transfer results. If three-dimensional effects are oversimplified in the model, analyses of the spatial results can be misleading and could cause inaccurate assessments. The missile body also contributes or enhances the three-dimensional effects that influence the plume size, inviscid shock structure, and plume shear layer growth. The body also appears to play an important role in the location of the barrel shock reflection point and the subsequent combustion processes in the far field (i.e., afterburning). The shock reflection point is moved substantially farther downstream, the combustion zone is sustained in the far field, and the plume radial extent is larger when the full 3D geometry is included in the computational domain.

The body influence in the plume far field was not anticipated. These simulations indicate that the missile body and base regions have a flameholding effect, intensifying and sustaining downstream combustion sources. The most dramatic influence for low-altitude LRE simulation is the flameholding effect captured by the missile body/plume simulation which was absent in the SEN simplifications.

Comparisons with in-band station radiation measurements confirm an axially extended, intense plume far-field radiation region. This phenomenon is attributed to a sustained plume/atmospheric afterburning condition. This effect was not represented in the model that did not incorporate the missile body and base in the computational domain. However, the comprehensive representation of the missile body/base captured the phenomena and predicted intensity trends which favorably compared with the observation.

Simplification of the geometry and chemical approximations incorporated into flow-field models that interface with radiative transfer formulations will lead to inaccurate representation of the IR radiative heating under certain conditions. Complete representation of the missile/plume geometries appears to be of major importance to accurate IR radiative transfer predictions. Finite-rate chemistry approximations also are very important if conditions are conducive to shear layer combustion.

Future development of the GPACT computer program, which was applied in this study, will focus on numerical approximations for two-phase flow, phase change, and turbulence modeling. A major focus of the future work will also concentrate on improving the computational efficiency and robustness of the algorithm.

Acknowledgment

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